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**SUBJECT:** Mission Planning Presentation  
at OMSF Management Council on  
February 5, 1969 - Case 310

**DATE:** March 4, 1969

**FROM:** V. S. Mummert

MEMORANDUM FOR FILE

The attached viewgraphs (with comments) were presented at the Executive Session of the OMSF Management Council on February 5, 1969. They contain results of the two week NASA study of post-Apollo lunar missions and constitute the mission planning portion of the "Post-Apollo Lunar Mission" presentation.

Contributions to the study and the presentation from KSC, MSC, MSFC, TRW Systems, and the Apollo Program Office are acknowledged.

*V S Mummert*

V. S. Mummert

2013-VSM-cjz

Attachments

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## VIEWGRAPH 1

Viewgraph 1 shows the scope of the study undertaken by the Mission Planning Team. As shown here, two classes of missions were analyzed: lunar landing missions and lunar orbital remote sensing missions. Under lunar landing missions, the Team was asked to examine specifically the four separate issues or goals that are listed here and to treat them on the listed priority:

- (1) precision landing to a pre-selected target; (2) landing on any point on the surface of the moon;
- (3) increased payload to the lunar surface; and
- (4) increased payload from the lunar surface. Each of these issues will be addressed during the course of this briefing but the briefing sequence will not follow this priority list. It should be pointed out that payload is here used in its broadest sense to include added consumables, added habitability and scientific equipment. Under the lunar orbit mission the Mission Planning Team examined the polar orbital mission for two reasons - first, it seems to be of greatest interest to the scientists and others who write the requirements for the mission, and secondly, it apparently is the most difficult mission to plan and on the whole has harder, more difficult abort problems. So being more difficult, it in some sense bounds the problem.

## MISSION PLANNING

- LUNAR LANDING MISSIONS
  - PRECISION LANDING TO A PRESELECTED TARGET
  - LANDING AT ANY POINT ON FRONT SURFACE OF MOON
  - INCREASED PAYLOAD TO THE LUNAR SURFACE
  - INCREASED PAYLOAD FROM LUNAR SURFACE
- LUNAR ORBITAL REMOTE SENSING MISSION
- POLAR MISSION

## VIEWGRAPH 2

The second viewgraph is a reminder that the goals set up for the lunar landing mission are not independent and the interdependence is illustrated with some tradeoff factors. First, 10 lbs. of payload to the surface is approximately equivalent to 12 lbs. from the surface, the limiting factor here being the capability inherent in the LM Ascent Propulsion System. This fact gives us a direct tradeoff between the last two factors in the lunar landing priority list. It also must be noted that there is a tradeoff between payload to the surface and accuracy, namely, 10 lbs. of payload to the surface is approximately equivalent to 150 feet of redesignation if the maneuver is made 30 seconds after Hi-Gate. And, as you know, redesignation is a primary means of terminal phase guidance. The final point reminds us that 10 lbs. of payload to the surface is approximately equal to 12 lbs. of CSM inert weight. So here is shown a possible direct tradeoff between the experiments landed on the surface and the experiments carried in the CSM.

## LUNAR LANDING GOALS

### NOT INDEPENDENT

- 10 LBS. PAYLOAD TO SURFACE  $\approx$  12 LBS FROM SURFACE

10 LBS. PAYLOAD TO SURFACE  $\approx$  3 FPS IN DESCENT  $\Delta V$

$\sim$  150 FT REDESIGNATION AT 30 SEC

$\approx$  .5 SEC HOVER

- 10 LBS. PAYLOAD TO SURFACE  $\approx$  20 LBS SEPARATION WT

$\approx$  8 LBS CSM INERT WT

### VIEWGRAPH 3

Addressing directly now the first point to be discussed under the lunar landing, the capability to land at any point on the moon's face, it is apparent that there is some problem in locating or synthesizing the most difficult point on which to carry out analysis. Hence, the question is changed to ask "Can we land on any designated science site on the moon's face". It will be shown that the capability exists in the CSM if less constrained trajectories are used. It is profitable to spend a few minutes describing these trajectories because they are a departure from Apollo's present way of doing business. In order to fly these less constrained trajectories, a 14 day CSM capability is presumed and it will also be necessary to redesign or requalify the LM Descent Propulsion System supercritical helium system to accommodate longer flight times. The longer flight times result from the lower energy trajectories in question. The Engineering Team will discuss specifically the hardware implications of a 14 day CSM and the revised supercritical helium system. Before going on, it should also be pointed out that there are no identifiable problems for supporting the missions in question here. MSC will be prepared to do the required pre-mission planning and also real-time mission control in the timeframe in question. Also, there are no identifiable targeting problems at MSFC.

LAND AT ANY POINT ON MOON'S FACE

- LAND AT ANY DESIGNATED SCIENCE SITE ON MOON'S FACE
- PERFORMANCE CAPABILITY EXISTS IN CSM
- BUT LESS CONSTRAINED TRAJECTORIES REQUIRED
- 14 DAY CSM CAPABILITY IS PRESUMED
- SUPPORT FOR THESE MISSIONS NOT A PROBLEM

#### VIEWGRAPHS 4 AND 5

The next two viewgraphs illustrate the kind of translunar trajectories that have been the subject of discussion. On the left (on Viewgraph 5) is the circumlunar free return trajectory which is currently being planned for the first few Apollo missions; it's familiar to all of you. It only assumes that an "abort" capability exists in the SM RCS system. On the right is a purely unrestricted non-free return trajectory which in the most general sense requires the availability of the SPS in order to effect a safe earth return. In the middle is illustrated the so-called hybrid trajectory which initially starts as a circumlunar free return (black line) having a high altitude perilune on the order of 100 to 5,000 miles. The high altitude perilune, of course, is equatable with low energy trajectories and part of the efficiency of the hybrid comes from this particular fact. At the point which has been labeled "go hybrid" the free return trajectory is converted to a non-free return trajectory (the red line) which is constrained by a requirement that it be possible to abort using the LM descent propulsion system at something like 2 hours after normal lunar orbital insertion (the blue line). This trajectory, the hybrid trajectory, has a virtue in that it starts out free return and is dependent only on the SPS working at lunar orbit insertion after it has worked at the hybrid midcourse point or on a DPS maneuver while in the docked configuration. It is also possible to identify, as has been done, a subclass of hybrid trajectories which are shaped without regard to any DPS abort constraint. They will have the safety feature of initially being on a free return but generally will be dependent on SPS operations beyond this point. As set out on Viewgraph 4, the translunar trajectory types are listed in generally descending order of safety and ascending order of SPS propellant efficiency.

Another feature which is expected to be used for lunar exploration missions is the so-called three-impulse Lunar Orbit Insertion (LOI) and Transearth Injection (TEI). These methods use intermediate parking orbits with a high apolune to minimize the cost of the plane change. Since they are just techniques for getting into and out of parking orbit, they should be thought of as an inner loop on the basic lunar trajectory options. The three-impulse technique is given further elaboration on backup Viewgraph 5b.

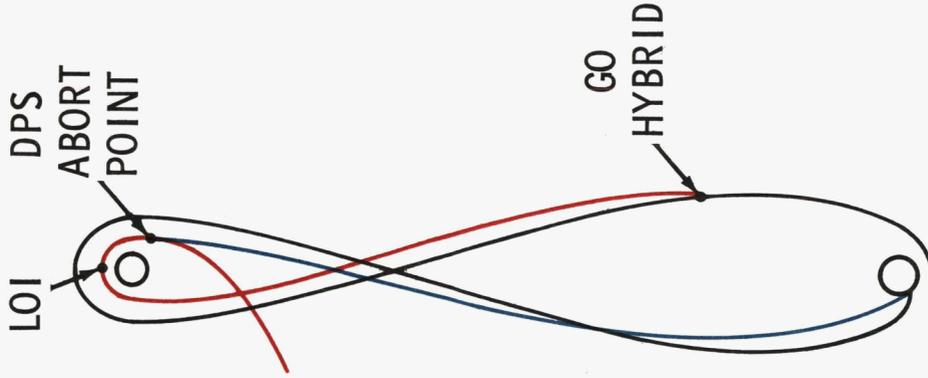
## TRAJECTORY OPTIONS

- TRANSLUNAR TRAJECTORY TYPES
  - FREE RETURN
  - HYBRID
  - WITHOUT DPS ABORT CONSTRAINT
  - NON-FREE RETURN
- THREE IMPULSE LOI AND TEI  
(INNER LOOP)

TRANSLUNAR TRAJECTORY OPTIONS



FREE RETURN



HYBRID



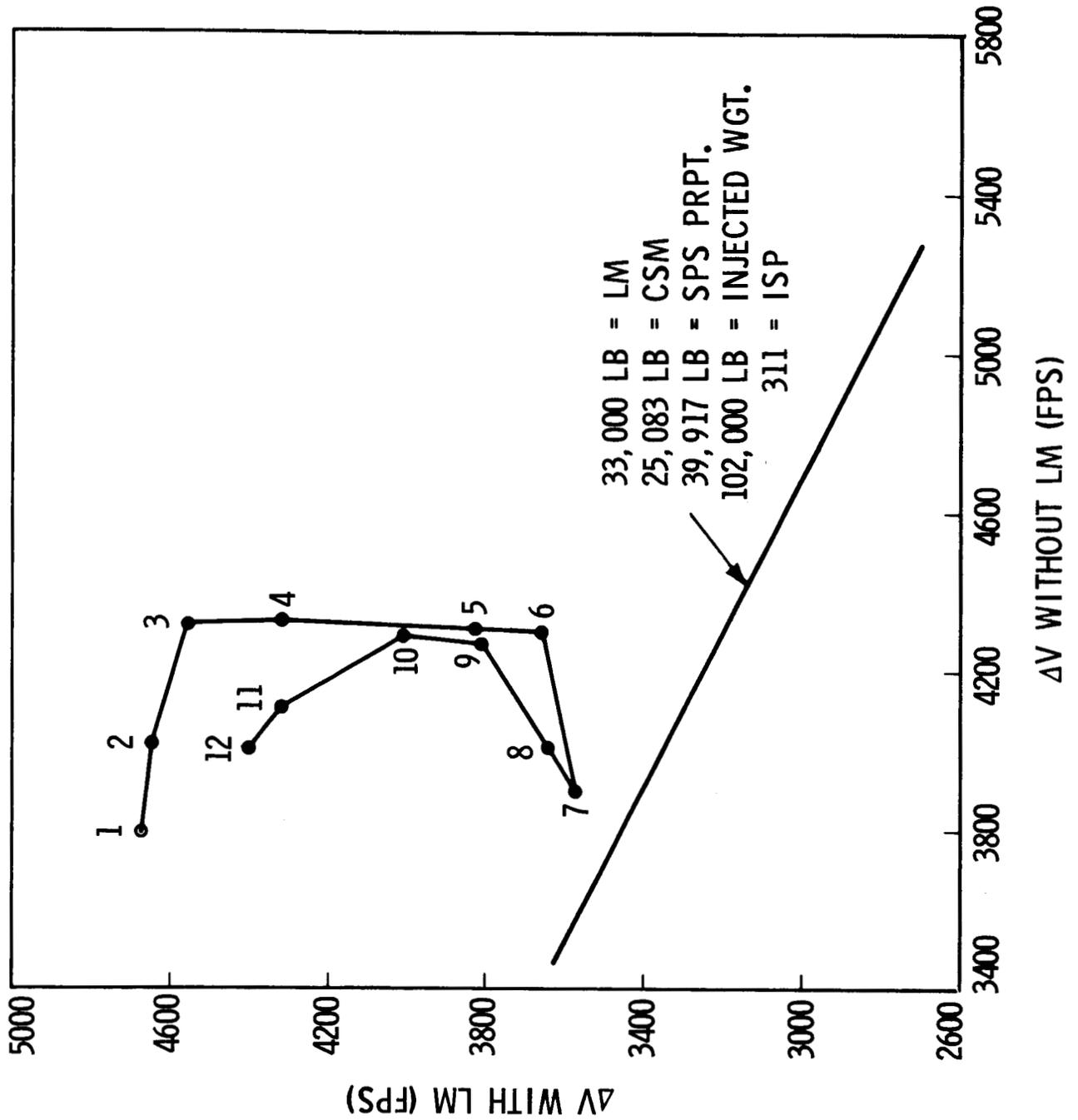
NON-FREE RETURN

## VIEWGRAPH 6

It is now appropriate to turn to a comparison of SPS performance requirements for the various translunar trajectory options which have been discussed. In doing this a typical science site, Littrow, located at about 22° N latitude and 29° E longitude, is chosen to make the point. Looking first at the broken, numbered line, it is seen that during 1971 there is one mission each month to Littrow if the lighting constraint currently applicable to the first lunar landing mission is imposed. Having this unique monthly mission provides two delta-V values, one for getting into parking orbit with the LM attached and one for the maneuvers in orbit and departure without the LM. One can then move through calendar year 1971 identifying a point in this delta-V plane for each month - one (January) through twelve (December) - and if we were to continue in 1972 we would see that the curve would essentially close on itself. So this locus then describes the requirements for getting to Littrow in 1971. The delta-V requirements plotted include the contingencies and dispersions identified for the first lunar landing.

The straight sloping line, on the other hand, describes the capability for a given vehicle configuration as indicated here. The vehicle in question is roughly equivalent to the current Apollo spacecraft with the exception that the CSM is about 1,000 lbs. heavier than the current weight. Any point on this capability line is equally within the capacity of the spacecraft in question. Points below the line have positive margins, points above the line have negative margins. It is immediately seen then that in no case do the requirements for getting to Littrow with free return trajectories fall within the capabilities of this vehicle.

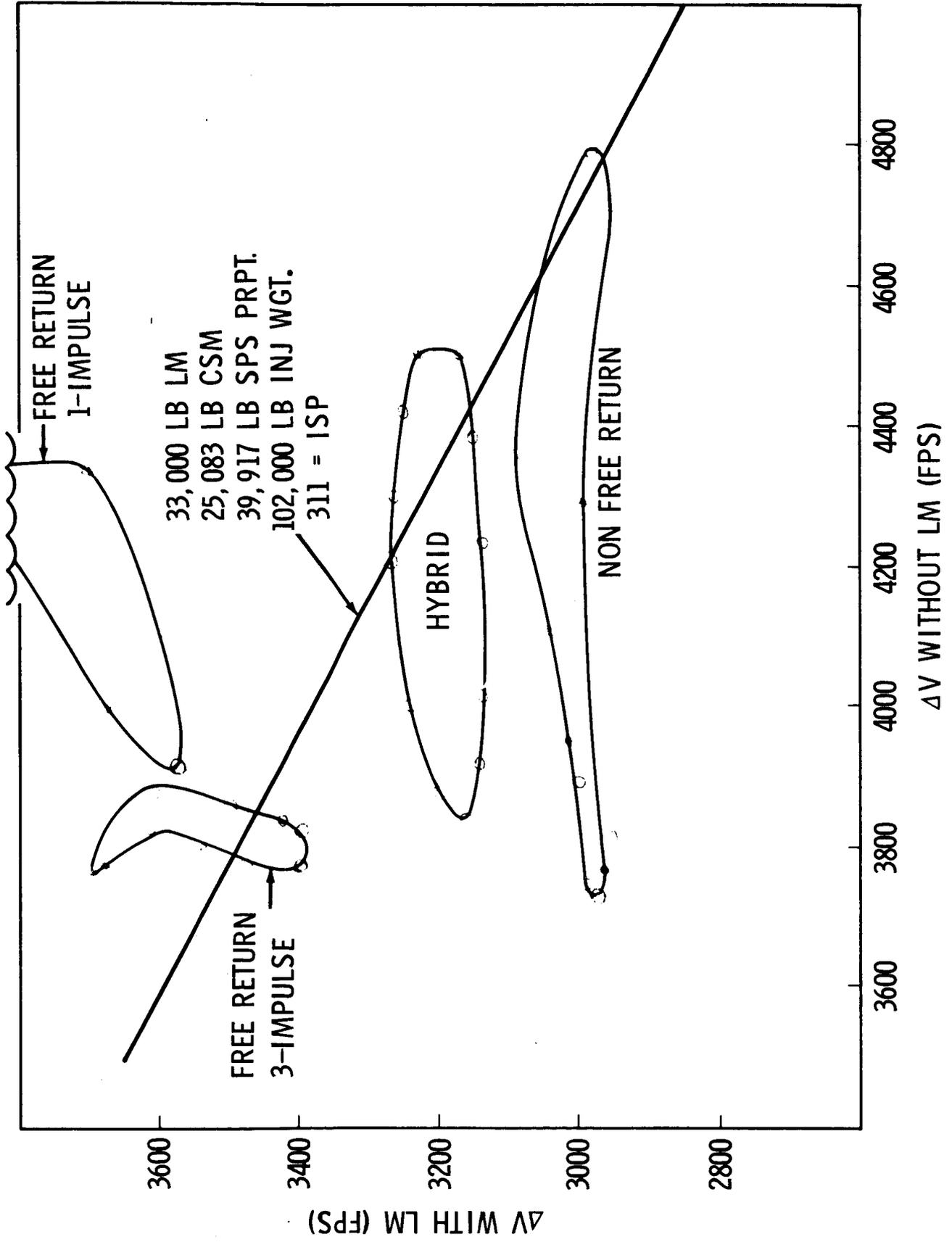
FREE RETURN  $\Delta V$  COSTS DURING 1971  
FOR LITROW (22°N, 29°E)



## VIEWGRAPH 7

Using the method developed on the last viewgraph now allows a comparison between the various trajectory options. On this viewgraph the scale has been changed so that the basic free return using one-impulse LOI and TEI which was previously discussed now is generally off the top of the chart and just three or four points fall on this chart. Free return trajectories using the three-impulse technique for LOI and TEI have about a quarter of their area under the line of constant capability. So it can roughly be said that Littrow would be accessible using free-return three-impulse techniques about three months out of the year. Moving to the hybrid trajectory now we see that almost every point lies under the capability curves so that we can get to Littrow using hybrids probably more than eight months of the year. Utilizing unrestricted non-free return trajectories, there is only one point, one month out of the year, that is beyond the capability of this spacecraft configuration.

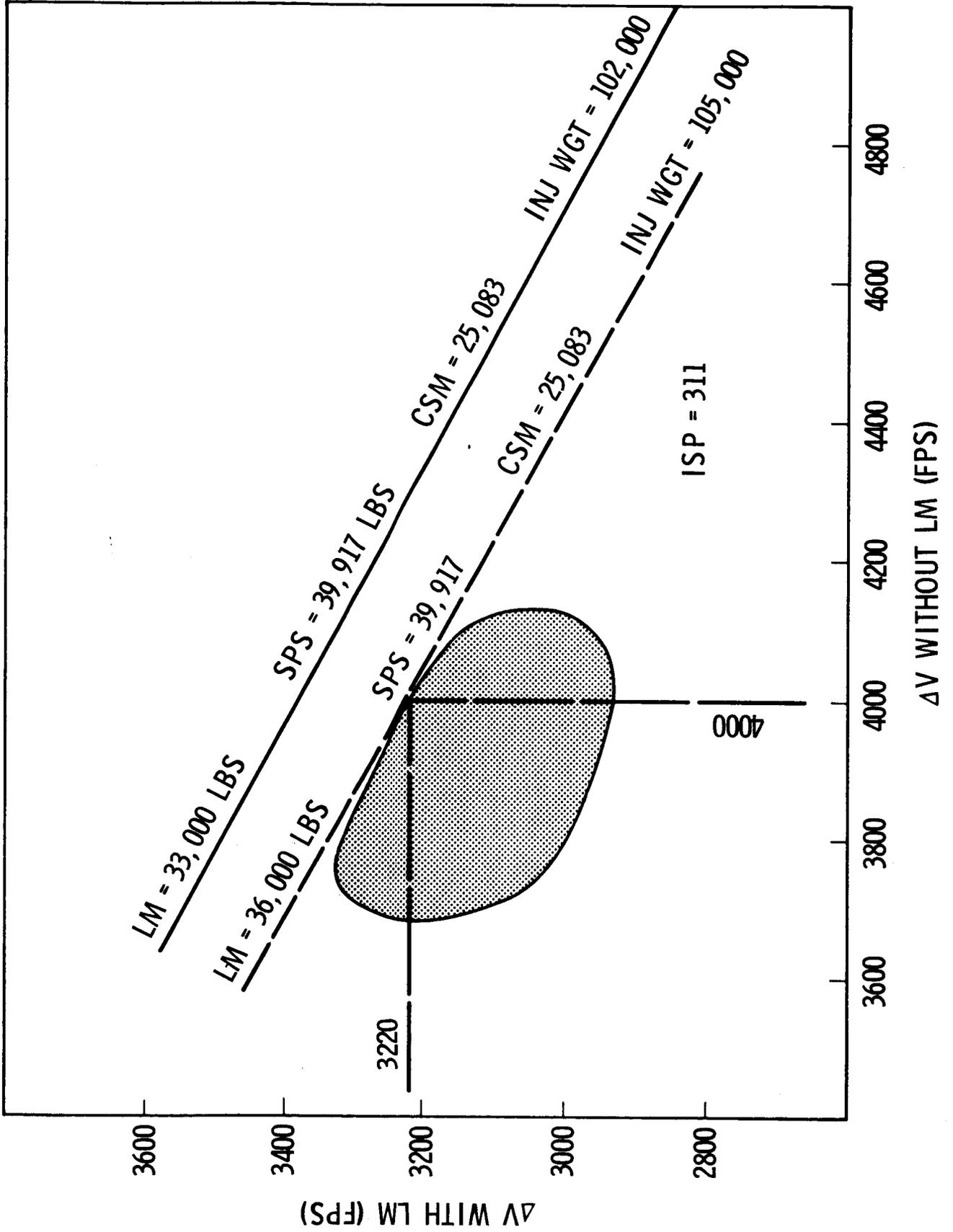
TRAJECTORY PROFILE COMPARISON  
FOR LITTRON (22°N, 29°E)



## VIEWGRAPH 8

In order to quantitatively evaluate the requirements for getting to all the designated science sites, a series of ground rules is required. First it is desirable to get to any given site on three consecutive months in order to allow some scheduling flexibility. We also allow the use of trajectories which go down through "hybrid without DPS abort constraint" on the list on Viewgraph 4. There then exists for each site three points which can be plotted in the delta-V plane. The sites examined generally include those described by Don Wise as being candidates for lunar exploration and in particular include Tycho, Aristarchus, and South of Alexander which are seen on backup Viewgraph 8b to generally bound the sites of interest. Having three points for each of the large number of sites which have been analyzed, an area as circumscribed on Viewgraph 8 is developed which is representative of the requirements for lunar exploration. A line drawn tangent to this area allows the calculation of an acceptable vehicle configuration; the point of tangency describes a delta-V budget as shown here, delta-V with the LM being 3220 fps and delta-V without the LM being 4000 fps. Notice that the configuration includes a LM weighing 36,000 lbs., clearly much larger than the current configuration.

LUNAR EXPLORATION  
 $\Delta V$  REQUIREMENTS



## VIEWGRAPH 9

Having looked at CSM capability and having found margins, it is now appropriate to turn to an examination of the LM capabilities. In particular, it is desirable to identify methods for transferring CSM margins to the LM and to examine the LM mission profile for possible margins to enhance either LM descent delta-V or landed weight. The first item, a lower lunar parking orbit, allows a possible saving of 110 fps if the planners are willing to lower the parking orbit altitude to 20 n.m. The 110 fps is also equivalent to 380 lbs. landed weight. It must be pointed out that all the items on this list have their disadvantages in that something must be given up in order to increase the delta-V margins. In particular, the lower lunar parking orbit incurs difficulties with aborts from LM descent in that phasing becomes more difficult. Navigation, as we currently understand it, may be more difficult because of the mass concentrations, and orbital sightings for navigation become more difficult as altitudes decrease. In fact, the 20 n.m. lower limit generally results from the limiting line-of-site rates during orbital navigation.

Engine modification to the descent propulsion system to allow high end throttling could result in delta-V savings on the order of 60 fps. The problems here are hardware changes with incumbent cost and schedule problems.

The Apollo system is now carrying a backup capability for lunar orbital rendezvous in the CSM. If part or all of the nominal rendezvous maneuvers were made with the CSM, enough ascent stage propellant could be saved to give the equivalent of up to 100 fps in descent delta-V or equivalently 350 lbs. of landed weight. This particular profile change has a unique built-in flight test feature in that it can be tested before it is needed. Both the CSM and the LM could be loaded for a nominal CSM active rendezvous with the LM as backup for one or two missions before the capability is actually required. Considerations for CSM active rendezvous are given further elaboration in backup Viewgraph 9b.

The last item on the list, flexibility, visibility, and redesignation requirements will be much better understood after the first lunar landing. For this reason we have listed the increment as being possibly plus or minus. It would be expected that rougher sites and the desired precision would cost more delta-V unless offset by margins now present in the delta-V budget. Because of a certain amount of uncertainty in this area, there is added incentive to develop added capability from the first three options.

POTENTIAL LM DESCENT  $\Delta V$ /LANDED WEIGHT INCREMENTS

LOWER LPO (20 NMI)	UP TO 110 FPS	OR	380 LBS
● HIGH END THROTTLING	UP TO 60 FPS	OR	200 LBS
● CSM ACTIVE RENDEZVOUS	UP TO 100 FPS	OR	350 LBS
● FLEXIBILITY, VISIBILITY, REDESIGNATION	PLUS OR MINUS		

## VIEWGRAPHS 10 AND 11

Turning now to the problems of precision landing to a preselected target (the first goal on the initial priority list), the next two viewgraphs set out the current problems in LM navigation and landing accuracy and compare the current capability with the requirements for lunar exploration. The list (Viewgraph 10) identifies a series of factors that contribute to landing dispersions and checks two of them as being the dominant contributors, namely, errors in orbit prediction due to the imperfect lunar potential model in the RTCC and errors due to propagation of the last navigation update in the LM onboard navigation system for about two orbits. The figure indicates the relative weight of the various items. Shown are 99 percentile lunar landing dispersion ellipses. The outer ellipse indicates the expected navigation uncertainties for an automatic landing on the G mission. It has a semi-major axis of about 30,000 feet. The next ellipse with a semi-major axis of about 15,000 feet is indicative of the dispersion which would be expected were we to introduce a perfect navigation update 4 hours before landing as on a current G mission flight plan. It can be said that the area between the outer two ellipses is representative of errors introduced by the RTCC lunar model. The third ellipse indicates the kind of accuracy that could be expected were the flight plan to be reconfigured so that a perfect update could be introduced at powered descent initiation (PDI).

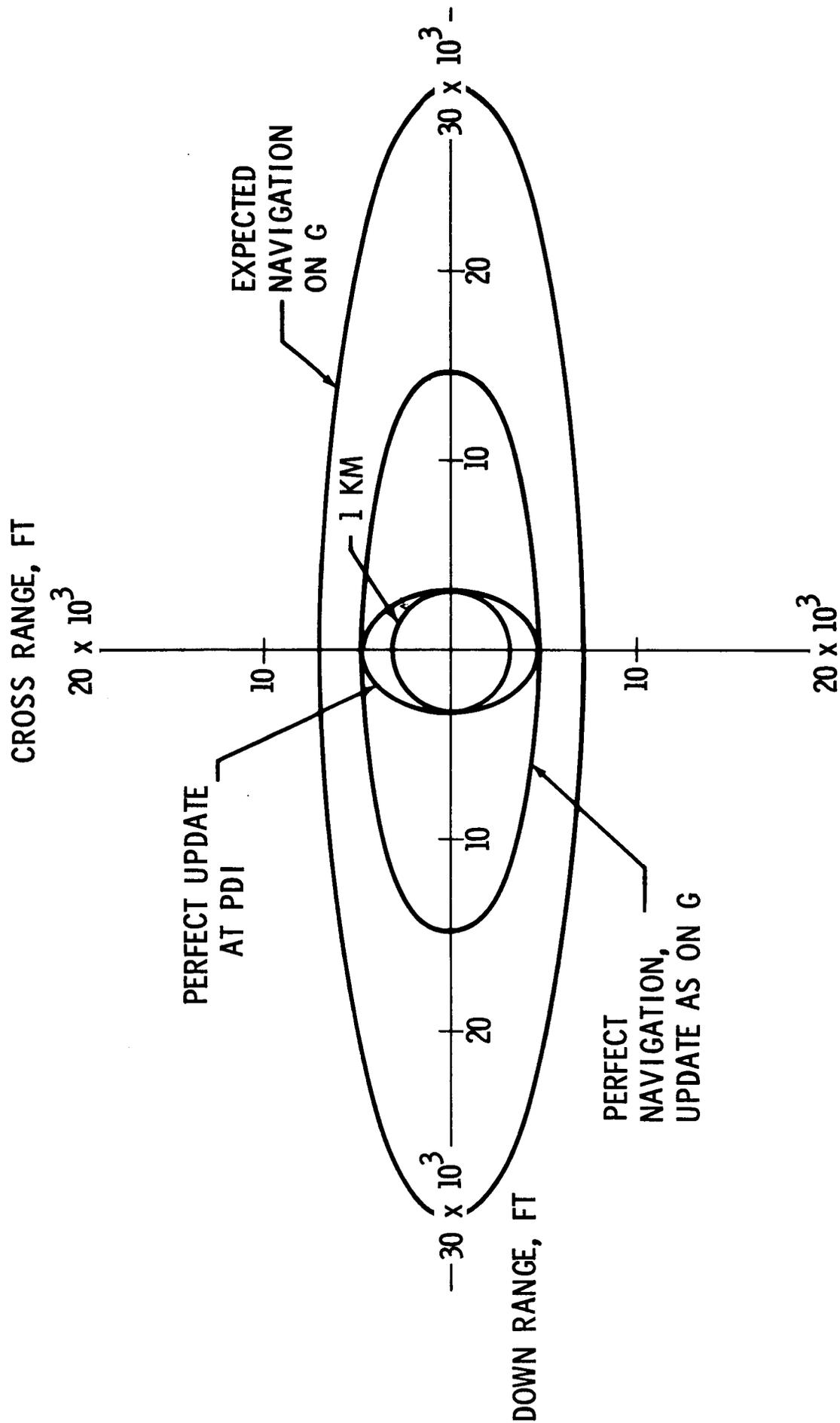
The area between the middle ellipse and small inner ellipse can be attributed to numerical approximations and drifts in the LM navigation systems which accumulate between update and powered descent initiation. Hence this area, about half the total error, is generally representative of dispersions which could be eliminated were the last update to be moved closer to ignition.

One problem is readily apparent. The present guidance scheme using the landing point designator and terminal phase guidance to reach the desired lunar landing site was never intended to steer out dispersions of the order of 30,000 feet or even 15,000 feet. The solution becomes one of improving navigation at powered descent initiation. Notice that the introduction of a perfect update at powered descent initiation gives accuracies which are readily comparable to the kind of accuracy, 1/2 kilometer to 1 kilometer, believed necessary for effective lunar exploration.

## CAUSES OF LANDING DISPERSION

- NAVIGATION
  - ✓ ● LUNAR MODEL (RTCC)
  - LANDMARK LOCATION
  - LANDMARK SIGHTING ACCURACY
  - MSFN
- FLIGHT PLAN
  - ✓ ● ERROR PROPAGATION FROM LAST STATE  
VECTOR UPDATE
- SYSTEMS
  - IMU ERRORS
  - LUNAR MODEL (AGC)

# LUNAR LANDING DISPERSION ELLIPSES (99%)



## VIEWGRAPH 12

Several auxiliary navigation techniques are being examined and are enumerated here. The first two would reduce the dispersion by putting the update later in the timeline. We could put it in one orbit before descent or perhaps introduce the update in the last orbit prior to descent. The latter technique is believed only feasible for sites west of about 15° longitude because of the time required to track, compute the orbit, complete and verify the update, and prepare for ignition.

The third technique received considerable attention from the Guidance and Control Division at MSC about a year ago but was abandoned because of problems of memory capacity in the LM guidance computer and because navigation was not thought to be the critical problem that it now is. This technique involved onboard LM sightings of a landmark uprange of the actual landing site. The position of this landmark and of the site would be calculated with respect to the orbit so that the relative positions were known and a mark on the landmark would be made on the pass just prior to ignition. The technique requires some sort of sighting technique using LM optics and added software to utilize the sighting data. The accuracy is limited by the accuracy to which the landmark position is known with respect to the site and by the accuracy of the sighting technique itself. But using this technique, orbital navigation accuracies would in theory approach that of the perfect update at PDI.

Another technique which has been considered involves triangulation by CSM sightings of the LM and a landmark near the landing site during the LM powered descent. It would require development of procedures with the CSM and a CSM-LM data link.

## AUXILIARY NAVIGATION TECHNIQUES BEING EXAMINED

- REDUCE TIME FROM MSFN UPDATE TO POWERED DESCENT
  - REDUCTION TO ONE ORBIT PROPAGATION
- MSFN UPDATE ON LAST PASS JUST PRIOR TO POWERED DESCENT
  - MAY BE FEASIBLE FOR SITES WEST OF 15° W LONGITUDE
- ON-BOARD LM SIGHTINGS OF UPRANGE LANDMARK
  - REQUIRES A LANDMARK SURVEYED RELATIVE TO THE DESIRED LANDING POINT
  - REQUIRES A SIGHTING TECHNIQUE WITH LM OPTICS (COAS, LPD, AOT)
  - ACCURACY REQUIREMENTS NOT STRINGENT
  - REQUIRES ADDED SOFTWARE TO UTILIZE SIGHTING DATA TO REDESIGNATE TARGET
- CSM TRIANGULATION DURING LM APPROACH OF LANDING SITE AND LM USING CSM OPTICS, RENDEZVOUS RADAR, VHF RANGING

### VIEWGRAPH 13

We now turn to a brief analysis of the Lunar Polar Orbit Reconnaissance Mission. It should be pointed out that mission analysis in this particular area is much less mature than the area of lunar landings. The basic mission involves the free return hybrid translunar trajectory with multiple impulse LOI and TEI and 14 days in the low lunar orbit, retention of an any orbit abort-to-earth capability and orbital plane changes as required to maintain the appropriate lighting conditions in the face of orbital procession. The analysis then indicates that launch opportunities exist every month with a science payload of about 14,000 lbs. to orbit as in a sub-satellite or in a LM ascent stage instrument carrier, or about 5200 lbs. could be accommodated round trip as in the Service Module empty bay.

## LUNAR POLAR ORBIT MISSION

- FREE-RETURN HYBRID TRANSLUNAR TRAJECTORY
- MULTIPLE IMPULSE LOI AND TEI
- 14 DAYS IN LOW ORBIT; 24 DAYS TOTAL MISSION TIME
- COVER 50% OF MOON IN SUNLIGHT
- COVER 50% OF MOON IN DARKNESS
- ANY ORBIT RETURN-TO-EARTH CAPABILITY
- ORBITAL PLANE CHANGES TO RETAIN LIGHTING
- LAUNCH OPPORTUNITIES EVERY MONTH
- MISSION PLANNING SCIENCE PAYLOAD

14,000 LBS TO ORBIT

OR

5,200 LBS ROUND TRIP

#### VIEWGRAPH 14

In summary then the LM landing accuracy is highly dependent on orbital navigation including the timeline on which the updates are made. This particular area requires the greatest program attention in the field of mission planning. It was shown that by using less constrained trajectories the CSM can potentially deliver a 36,000 lb. LM to designated science sites and SM performance does not seem to be a problem. Potential LM landed weight increases of up to 930 lbs. are possible through changes in the LM descent flight profile. Equivalently, these profile changes would make available 270 fps more for LM descent delta-V. Finally, preliminary orbital mission planning indicates that high performance margins exist in the CSM for the anticipated payloads required.

## MISSION PLANNING SUMMARY

- LM LANDING ACCURACY HIGHLY DEPENDENT ON ORBITAL NAVIGATION. AREA REQUIRES PROGRAM ATTENTION
- BY USING LESS CONSTRAINED TRAJECTORIES, CSM CAN DELIVER 36,000 LB LM TO DESIGNATED SCIENCE SITES
- POTENTIAL LANDED LM WEIGHT INCREASES OF UP TO 930 LBS EXIST
- PRELIMINARY ORBITAL MISSION PLANNING INDICATES HIGH MARGINS

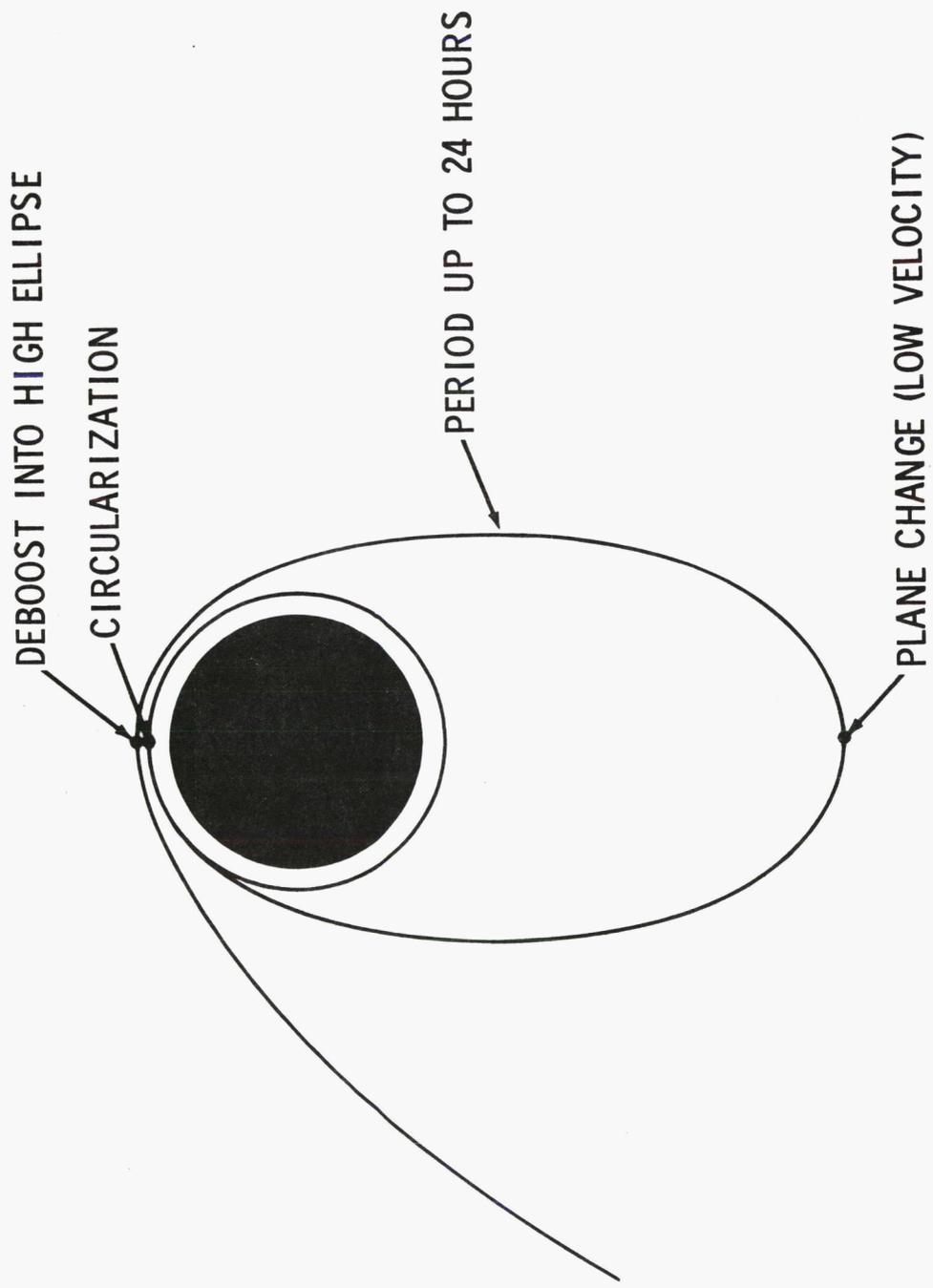
B A C K U P   V I E W G R A P H S

## VIEWGRAPH 5b

This viewgraph illustrates schematically a polar view of the so-called three-impulse lunar orbit insertion (LOI). Beginning on the left on the approach hyperbola, the vehicle is first deboosted into an intermediate elliptical parking orbit having a period of up to 24 hours. Some plane changes may be made during the deboost maneuver but the primary intent is energy reduction. Near apolune of this elliptical orbit, the primary plane change is made at a point of low velocity. The vehicle then continues to perilune where circularization takes place. The efficiency of this technique derives through making the plane change at a point of low velocity. The period of the intermediate ellipse (altitude of apolune) is a tradeoff between propellant optimization and total mission duration.

The three-impulse TEI follows the same general flight plan with a reverse sequence of events.

### 3 - IMPULSE LOI



VIEWGRAPH 9b

As proposed here, CSM active rendezvous would involve placing the CSM in a 20 n.m. altitude lunar parking and inserting the LM into a 30 n.m. orbit. The CSM active rendezvous would then be very similar to the current Concentric Flight Plan with a delta-h of 10 n.m. and rendezvous from below and behind.

The primary motivation for introducing a CSM active rendezvous in the nominal flight plan is the budget saving of 300 fps in LM ascent delta-V. As shown this is equivalent to 350 lbs. in surface payload or 100 fps in LM descent delta-V. The CSM delta-V budget, of course, now contains sufficient allocations for the maneuvers and the procedures are available for the contingency situation. As has been pointed out this mode can be efficiently flight tested before it is needed by maintaining a capability on the LM as well as the CSM and using the LM as a backup during the test phase. Possible techniques for nominal CSM active rendezvous are set out along with the apparent changes to the present system required to implement them. The data link now appears to be the most satisfactory solution if this mode is made nominal. It must be noted that the CSM rendezvous requires considerable RCS propellant and as such reduces the propellant available for attitude control during orbital science operations. Analysis has indicated that there are no unmanageable thermal problems for a nominal CSM parking orbit at 20 n.m.

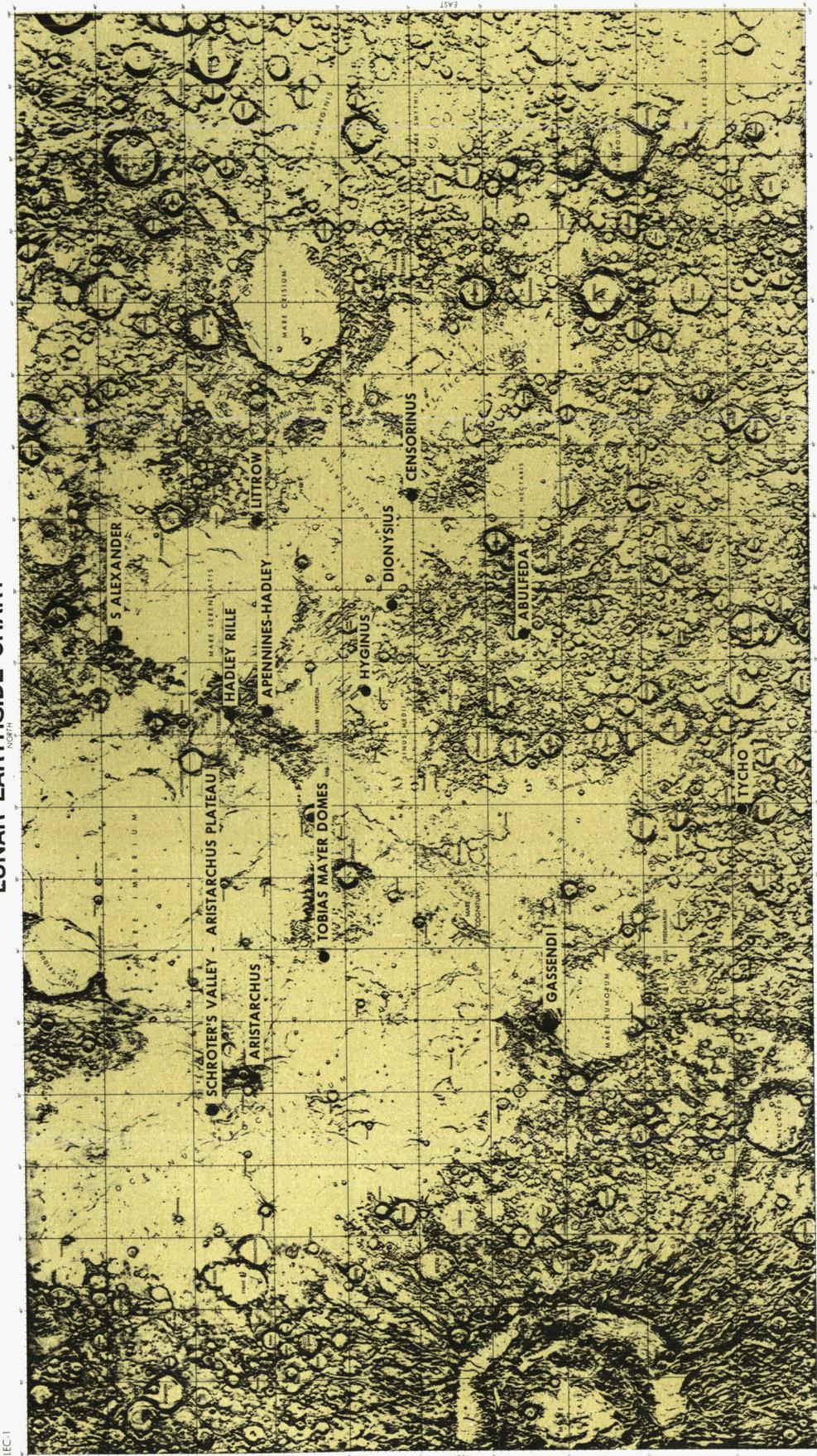
## CSM ACTIVE RENDEZVOUS

- SAVE 300 FPS ON LM RCS
  - = 350 LBS ON SURFACE
  - = 100 FPS ON LM DESCENT
- CSM BUDGET NOW CONTAINS ALLOCATION FOR MANEUVERS
- MODE CAN BE EFFICIENTLY " FLIGHT TESTED " BEFORE NEEDED
- POSSIBLE TECHNIQUES
  - PRESENT RESCUE MODE UPDATED TO 30 MI. LPO (PROCEDURES)
  - ADD RANGE RATE TO VHF (HARDWARE)
  - VOICE LINK RR DATA TO CSM (PROCEDURES)
  - DATA LINK FOR RR DATA TO CSM (HARDWARE AND SOFTWARE)
  - ADD RR TO CSM (HARDWARE AND SOFTWARE)
- CSM ACTIVE RENDEZVOUS REDUCES SM-RCS FOR ORBITAL SCIENCE
- NO UNMANAGEABLE THERMAL PROBLEMS AT 20 MI.

VIEWGRAPH 10b

This viewgraph illustrates the number and position of lunar sites which have been analyzed for SPS propellant costs on lunar exploration missions.

# LUNAR EARTH-SIDE CHART



IEC-1

LUNAR EARTH-SIDE CHART (IEC-1)  
SCALE 1:5,000,000  
PROVISIONAL



NOTE: The information on this chart is based on the data from the Lunar Orbiter spacecraft. The information is preliminary and is subject to change as more data becomes available. The information is based on the data from the Lunar Orbiter spacecraft. The information is preliminary and is subject to change as more data becomes available. The information is based on the data from the Lunar Orbiter spacecraft. The information is preliminary and is subject to change as more data becomes available.

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